100-LBF LO_2/LCH_4 REACTION CONTROL ENGINE TECHNOLOGY DEVELOPMENT FOR FUTURE SPACE VEHICLES

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ABSTRACT

The National Aeronautics and Space Administration (NASA) has identified liquid oxygen (LO2)/liquid methane (LCH4) propulsion systems as promising options for some future space vehicles. NASA issued a contract to Aerojet to develop a 100-lbf (445 N) LO2/LCH4 Reaction Control Engine (RCE) aimed at reducing the risk of utilizing a cryogenic reaction control system (RCS) on a space vehicle. Aerojet utilized innovative design solutions to develop an RCE that can ignite reliably over a broad range of inlet temperatures, perform short minimum impulse bits (MIB) at small electrical pulse widths (EPW), and produce excellent specific impulse (Isp) across a range of engine mixture ratios (MR). These design innovations also provide a start transient with a benign MR, ensuring good thrust chamber compatibility and long life. In addition, this RCE can successfully operate at MRs associated with main engines, enabling the RCE to provide emergency backup propulsion to minimize vehicle propellant load and overall system mass.

INTRODUCTION

NASA initiated technology development projects for a LO2/liquid hydrogen (LH2) auxiliary propulsion system (APS) in 1971 during the Space Shuttle pre-development period (Ref. 1-4). Since 1971, Aerojet has been supporting NASA's continuing efforts to develop alternatives to hypergolic propellants for APS applications. Specifically, Aerojet has leveraged projects funded by NASA, the United States Air Force (USAF), and Aerojet Independent Research and Development (IR&D) to expand the non-hypergolic propellant technology database for

space vehicle applications. While early work focused on LO2/LH2, the majority of these projects over the past 35+ years have focused on extending the technologies for oxygen/ hydrocarbon propellant combinations (Ref. 5). Aerojet has been involved with the development of a variety of combustion devices and propulsion systems that utilize oxygen and a range of potential hydrocarbon fuels, including RP-1, RP-2, JP-10, propane, ethanol, and methane (Ref. 6-12). This background has proven beneficial with respect to NASA's interest in advancing technologies for oxygen/hydrocarbon propellant systems.

The NASA Exploration Systems Architecture Studies (ESAS), along with multiple other study activities, have identified that LO2/LCH4 propulsion systems are a promising option for some future space vehicles. Specifically, an integrated main and reaction control propulsion system utilizing LO2/LCH4 propellants can provide substantial savings in overall systems mass when compared to conventional hypergolic systems.

This LO2/LCH4 propulsion system utilizes a common pressurization system, common propellant tankage, and a sub-cooled liquid feed system with pressure-fed reaction control system (RCS) thrusters and a single or multiple pressure-fed main engine(s). Although recent work in LO2/LCH4 technology advancement has shown progress, this propellant combination is still considered novel and requires a dedicated risk reduction program prior to proceeding with detailed design, development, and fabrication of an integrated LO2/LCH4 propulsion system. To address this necessary risk reduction, the NASA Exploration Technology Development Program (ETDP) has formed a project entitled Propulsion and Cryogenics Advanced Development (PCAD). The PCAD project is jointly managed by the NASA Glenn Research Center (GRC) in Cleveland, Ohio, and the NASA Johnson Space Center (JSC) in Houston, Texas.

The charter of PCAD is to identify and fund programs that develop and expand the maturity of candidate technologies considered to be important for future cryogenic space vehicles. These programs focus on components or subsystems that are deemed lacking in technical maturity but are considered to be essential to successful design, development, and fabrication of an integrated LO2/LCH4 propulsion system. Consistent with PCAD's charter, NASA issued a contract to Aerojet to develop a 100-lbf LO2/LCH4 RCE aimed at reducing the risk of utilizing a cryogenic RCS on a space vehicle.

RCE PROGRAM OVERVIEW

The NASA-sponsored 100-lbf RCE technology development program was divided into four phases: Basic, Option 1, Option 2, and Option 3. The Aerojet program plan for this development effort was structured around these four major phases.

During the Basic Phase, a flight-type detail design was produced that incorporated the features necessary to meet the established operational and design requirements for the engine. The Basic Phase concluded with a critical design review (CDR), and generated a complete design disclosure with supporting analyses to enable fabrication of the engine.

Option 1 then took the completed design disclosure from the Basic Phase and fabricated RCE hardware for hot-fire testing. Two different injector and two different igniter designs were fabricated and tested during Option 1. Option 2 provided the opportunity to perform a design iteration based on the results of the Option 1 testing, and Option 3 fabricated four engines to be delivered to the NASA White Sands Test Facility (WSTF) to be incorporated into a cryogenic APS Test Bed (APSTB). Testing with the cryogenic APSTB provided the opportunity to evaluate an APS with cryogenic fluid management at simulated altitude conditions.

The purpose of this paper is to highlight some of the more important results from the Basic and Option 1 phases of the 100-lbf LO2/LCH4 RCE program.

RCE DESIGN DESCRIPTION

The Basic Phase of the program established the NASA/Aerojet conceptual flight-type RCE design. This conceptual design enabled the RCE development challenges to be identified along with the corresponding risk reduction activities necessary to address these challenges. The five major technical challenges identified and resolved by the NASA/Aerojet team for the RCE are:

(1) Oxygen-rich (high MR) start transients and potential chamber burn-through
(2) Reliable and repeatable ignition over a large range of valve inlet temperatures (liquid-liquid to gas-gas conditions)
(3) High engine performance (Isp)
(4) Low MIB and pulse-to-pulse repeatability
(5) Operation of the RCE at an MR above 3.0 to provide main engine redundancy during flight operations.

The NASA/Aerojet flight-type RCE design consists of the following components: a compact integral exciter/spark plug system, a dual coil direct-acting solenoid valve for oxidizer and fuel, an integral igniter and injector, and a columbium chamber/nozzle with an expansion area ratio of 80:1. The Option 3 engines delivered for WSTF APSTB testing are identical to the flight-type engine with two exceptions: the exciter is not a compact integral design and the nozzle expansion area ratio is 45:1. Figure 1 shows the Option 3 engine.

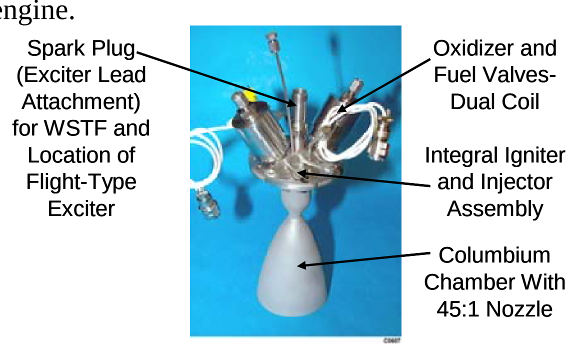

Fig. 1: Deliverable Option 3 White Sands Flight-Type RCE

The 100-lbf LO2/LCH4 RCE program has provided risk reduction design/hot-fire data on all five critical challenges and, where permitted, this data will be provided to demonstrate verification of the design requirements/technical challenges.

The exciter used for hot-fire testing is a high-voltage capacitive-discharge-type unit that is used in conjunction with a spark plug to provide an ignition source for the igniter. This exciter, Figure 2, utilizes a high-voltage lead running from the exciter box to the spark plug, and is similar to previous flight qualified units such as the RL-10 exciter. The high-voltage lead of the current exciter presents a significant reliability risk due to the potential for the corona discharge phenomenon occurring at very low external pressures, that is, high altitude or space. This corona discharge phenomenon bleeds off the high voltage intended for the spark plug through poorly isolated or damaged areas of the lead, thus delivering either a weak spark or no spark at all, causing non-ignition of the igniter and engine.

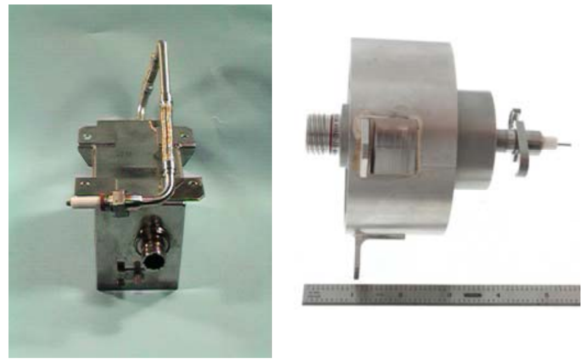

Fig. 2: Current Exciter Used for WSTF Altitude Testing and Integral Exciter in Development

A new exciter design is being developed in parallel with the RCE that overcomes the corona discharge problem described above. This design is based on trade studies identifying the ideal configuration as one having an integral excitation system and spark plug, thus eliminating the high-voltage lead. Therefore, the new exciter design, shown in Figure 2, improves reliability and safety of the unit, in conjunction with a smaller volume and mass.

The propellant valves are direct-acting solenoid-type designs, with the fuel and oxidizer valves being of similar design and materials. The solenoid of each valve contains dual coils for redundancy, enhancing the operational reliability of the engine assembly. The valves mount directly to the back of the injector by means of threaded outlet ports, thereby minimizing the injector manifold dribble volumes for each propellant circuit.

The RCE utilizes a novel injector design that integrates the igniter and the injector into one component. This integrated igniter/injector eliminates the need for a separate set of propellant valves for the igniter, thus reducing engine mass and volume. The integral igniter/injector is constructed from nickel and nickel alloy materials for oxygen compatibility and manufactured by Aerojet using a proprietary platelet fabrication process. The Aerojet platelet process photo-

chemically machines (PCM) very thin plates (platelets) that are subsequently stacked on top of each other and fused into a monolithic part during a diffusion bonding process at temperature and pressure. This platelet process enables small intricate flow passages to be incorporated into an injector design to promote uniform fluid distribution at the injector face.

The chamber/nozzle is fabricated from a columbium alloy with a high temperature capability, and coated with di-silicide for oxidation resistance. The conceptual flight-type engine nozzle is an 80% Bell with an expansion area ratio of 80:1 to meet the performance requirements within the engine envelope constraints. The nozzles on the WSTF deliverable engines (Option 3) have an 81% Bell and area ratio of 45:1, and the nozzles utilized for sea-level testing at Aerojet (Option 1) have a 15-deg half angle conical nozzle with an area ratio of 3:1.

Mass is always a key requirement for any propulsion system component and the flight-type RCE design has a mass of just 11 lbm (5 kg). A comparable 100-lbf (445 N) earth storable RCE used on the Apollo Command Module has a mass of approximately 9 lbm (4.1 kg). The other important design and operational requirements/ parameters defined during the Basic Phase of the program are provided in Table 1.

Parameter	Specification Value
Thrust	100 lbf (445 N)
Chamber Pressure	175 psia (12 Bar) [Derived from 325 psia (22.3 Bar) Inlet Limit]
Specific Impulse	>317 sec
Mixture Ratio	2.6-3.5
Nozzle Area Ratio (% Bell)	Engine Envelope – 7-in. Dia by 16-in. Length (17.8 cm x 40.6 cm)
EPW – Minimum	$\leq$ 80 msec
Impulse Bit – Minimum	4 lbf-sec (17.8 N-sec)
Valve Cycle Life	25,000 cycles
Mass	Minimize [R4D Storable 100-lbf Engine ~ 9 lbm (4.1 kg)]

Table 1: LO2 LCH4 RCE Major Engine Requirements Defined in Basic Phase

RCE OPTION 1 TEST RESULTS

Option 1 utilized the design disclosure developed during the Basic Phase to fabricate four different integral igniter/injector assemblies. These four assemblies represent the combinations of two different injector face patterns and two different igniter designs. The injectors and igniters were referred to as Injector 1 and Injector 2 and Igniter A and Igniter B, respectively. These designs included all applicable key features of the flight-type igniter and injector designs so that these features could be evaluated during hot-fire testing. The primary objectives of the Option 1 test effort were to:

(1) Characterize MR start transient response and engine operating characteristics
(2) Determine igniter inlet operational range and repeatability
(3) Determine steady-state performance and the effect of varying Pc and MR
(4) Characterize engine pulse mode response and repeatability, MIB
(5) Establish high MR operational capability.

Option 1 testing was conducted at Aerojet's A-Zone Research and Development Test Facility in Sacramento, California. The NASA/Aerojet team conducted over 1450 sec of cumulative burn time and accumulated more than 1400 pulses in the 135 total tests performed. The A-6 Test Cell is uniquely configured with LN2 jacketed propellant run lines to provide a wide range of inlet temperatures to the engine valves. In addition, the test cell includes redundant thrust measurement and redundant flow measurement to enable calculation of Isp while minimizing instrumentation error. The Option 1 hot-fire engine test article, Figure 3, is identical to the WSTF deliverable unit except it uses a conical nozzle with a 3:1 exit area ratio to accommodate testing at sea level. The 3:1 nozzle allows extrapolation of Isp to higher area ratio nozzles. The Option 1 hot-fire test data provided validation of the major engine requirements as shown in Table 2.

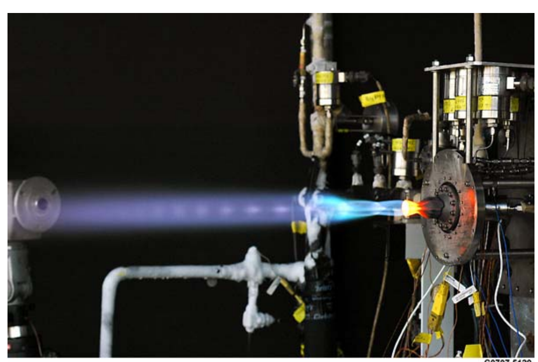

Fig. 3: Sea-Level Testing of Aerojet's LO2/LCH4 100-lbf RCE

Parameter	Specification Value	Option 1 Test Results
Thrust	100 lbf (445 N)	84 to 115 lbf (374 to 512 N)
Chamber Pressure	175 psia (12 Bar)	163 to 210 psia (11.2 to 14.4 Bar)
Isp	>317 sec	>320 sec (1) 317 sec (2)
Mixture Ratio	2.6 to 3.5	2.3 to 3.5
Nozzle Area Ratio, % Bell	7 by 16 in. Envelope	80:1 80% Bell Inside Envelope
EPW, Minimum	≤ 80 msec	40 msec
Impulse Bit, Minimum	4 lbf-sec (17.8 N-sec)	≤ 4 lbf-sec (≤ 17.8 N-sec)
Valve Cycle Life	25,000 Cycles	55,000 Cryogenic Cycles
Mass	Minimize	11 lbm With Flight-Type Components (5 kg)

Table 2: Option 1 Hot-Fire Results Validate Major Engine Requirements

MR Start Transient

The inability to predict the two-phase flow response in a dual cryogenic feed system is one of the biggest uncertainties in the development of a cryogenic RCS. A RCE design that is provided cryogenic liquids at the valve inlets must consider a high MR start transient as one of the most significant development risks. The NASA/Aerojet team chose to manage this risk with the use of innovative design features rather than using a control valve with active MR feedback. Such a valve would have added mechanical complexity and electrical impacts to the design, increasing system complexity, mass, and cost. Alternatively, gasification of the liquid propellants accompanied by separate storage for the RCS application would likewise increase system complexity, mass, and cost.

Another important characteristic when considering the start transient is the advantage of having simultaneous valve opening commands. A vehicle's guidance and control system provides electrical commands at some established internal clock speed, for example, every 30 to 40 milliseconds (msec). If the RCE start transient requires an oxidizer or fuel valve lead that is shorter than the clock speed, then a separate valve driver circuit is needed, further complicating the electrical design and mass of the engine assembly. The system benefit of using simultaneous valve opening commands simplifies the spacecraft switches and control logic necessary to manage the start transient. Aerojet recognized the system benefit of simultaneous valve commands and evaluated whether a passive control for the engine MR start transient could be achieved, which, if successful, would greatly simplify the system design and provide greater engine reliability.

Aerojet's approach to passive control for managing the start transient utilized the unique capabilities of their proprietary platelet process. This process enables creation of unique and specialized features within the igniter, injector, and propellant manifold passages. An example of one of these specialized features was in effect providing vacuum jacketed isolation around the main propellant fuel manifold to enhance rapid chill-in of that manifold. The ability to vacuum jacket the manifold is a by-product of diffusion bonding the platelets in a vacuum furnace.

In addition to specialized platelet features, the platelet process provides the versatility of adjusting the injector manifold internal design to ensure the fuel manifold chills in at the same time or ahead of the oxidizer manifold, thereby mitigating the risk of a high MR start transient. Option 1 testing verified that Aerojet's innovative passive control could manage the start transient as the MR during the start transient was limited to approximately 2.5 when the run tank pressures were set for a steady-state MR of 3.5. The tank set pressures for Option 1 Test 225 in Figure 4 were 350 psia (24.0 Bar) for the LO2 and 300 psia (20.5 Bar) for the LCH4.

Figure 4 documents the engine start transient for Injector 2. In the first graph, the engine chamber pressure (red) and engine MR (blue) are plotted for the 60-sec test. The plotted parameters clearly show the rapid rise of Pc to about 173 psia (11.8 Bar) at about 2 sec but the initial MR is about 2.5 in the same time period. The MR slowly increases from 2.5 to its steady-state value of 3.5. It takes approximately 1.0 sec for the flow meters to become steady; hence, the MR value below 1 sec is erroneous. The second graph of Figure 4 shows the injector manifold temperatures during the first 3 seconds of the test. The fuel manifold

temperature (red) crosses over and leads the oxidizer manifold temperature (green) during the first 250 msec, indicating fuel is chilling in ahead of oxidizer. Therefore, the unique manifold design features do produce a rapid fuel chill, as intended.

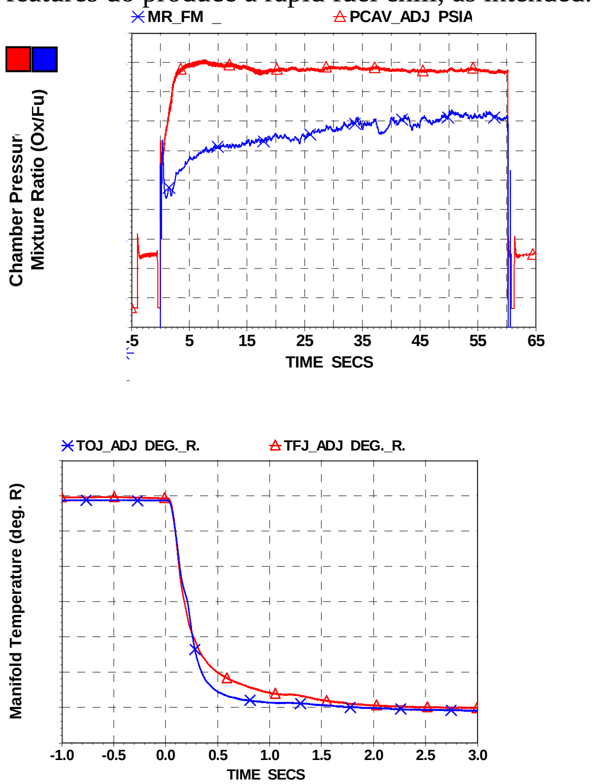

Fig. 4: MR Start Transient Mitigated by Aerojet's Unique Injector Manifold Design

Igniter Reliability and Repeatability

Aerojet's initial igniter development began in 1971 for gaseous oxygen (GO2)/gaseous hydrogen (GH2) propellants to support an early candidate cryogenic Space Shuttle APS under Contract NAS 3-14348 (Ref. 1). This Shuttle APS igniter evolved into the Integrated Thruster Assembly (ITA) in 1973 (Ref. 3). The igniter design was further refined and tested with GO_2/GCH_4 RCS thrusters for the X-33 program in 1997 (Ref. 10), which were flight qualified to Technical Readiness Level (TRL) 8. While these igniters worked well for gas-gas conditions, Aerojet conducted an IR&D ignition characterization program to define the ignition curve or quench parameter for the LO2/LCH4 propellant combination. The quench parameter determines the ignition limit below which combustion cannot be sustained within the igniter because energy is lost to the igniter walls at a faster rate than it is being generated. The quench parameter is critical in determining igniter orifice sizes, which is established empirically through ignition characterization testing.

In addition to empirically establishing the quench parameter for determining igniter orifice sizes, the igniter was integrated within the injector. The LO2/LCH4 100-lbf RCE is the first non-toxic RCS engine produced by Aerojet without a separate set of igniter valves to control igniter flow rates. Forgoing the separate set of igniter valves greatly simplifies the engine design and integration within the feed system, and lowers overall engine mass.

Aerojet incorporated three unique design features when integrating the igniter within the injector to ensure ignition within the igniter prior to the main chamber filling with propellant. In addition, the igniter design incorporated the ability to adjust the igniter MR with respect to the main injector. These features provided quick ignition, achieving nominal ignition times during Option 1 testing of 12 to 18 msec from valve opening command on Injectors 1 and 2 when utilizing Igniter A. Igniter A was verified to be superior to Igniter B, and never failed to ignite during the Option 1 test series at specified operational limits. Figure 5 shows the rapid response of the igniter chamber pressure (PCIGN) and main chamber pressure (PCAV) following the electrical command to the valves, as indicated by the propellant valve current traces (IOV, IFV). Ignition occurred in approximately 15 msec in this test.

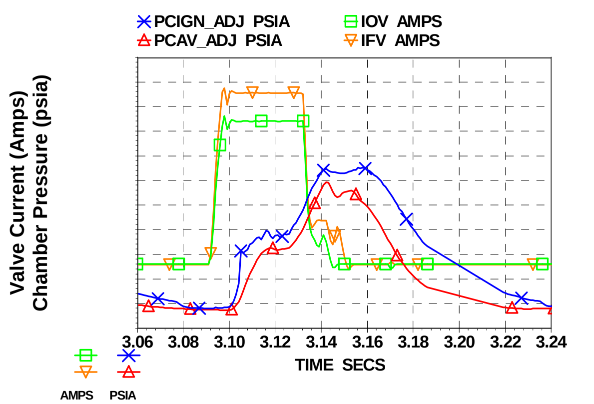

Fig. 5: Rapid Igniter and Main Chamber Pressure Response for Test 153 of Injector 1

Injectors 1 and 2 with Igniter A were tested over a broad range of inlet conditions to establish the

inlet temperature range over which ignition would occur. The widest possible range of inlet temperatures is preferred to provide the best system operability capabilities. Figure 6 shows the map of the inlet temperature range established by the Option 1 testing, and it clearly shows the igniter ignition range well beyond the specified operational limits. Igniter ignition occurred for all but one test during Option 1, whether steady-state or pulse tests, with the one non-ignition occurring at TOV of 240 R and TFV of 282 R during a pulse test (Test 177), which was well outside the specified operational requirements. The controlling propellant for ignition is the oxidizer to ensure an ignitable mixture ratio in the igniter.

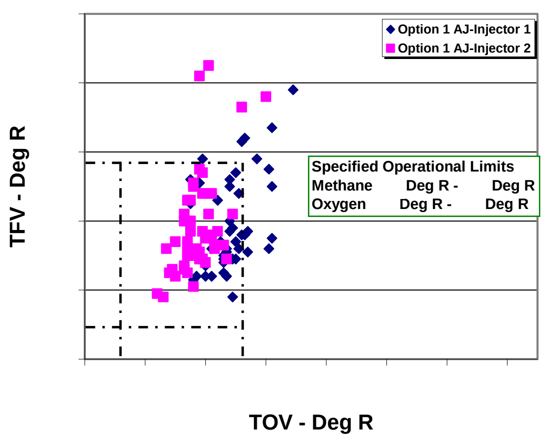

Fig. 6: Inlet Temperature Map of Igniter Ignition Points Demonstrating Broad Ignition Range

Isp Performance

Overall performance of the engine is governed by many factors that are dependent on the selected injection scheme but the most influential factors for the 100-lbf RCE are as follows: the momentum ratio between the fuel and oxidizer orifices, the element-to-element mixing efficiency, and the MR that is dependent on the propellant temperature/density. The RCE test configuration used redundant flow measurement and redundant thrust measurement to minimize the effects of uncertainty for calculating sea-level Isp for the 3:1 nozzles. The calculated 3:1 Isp was corrected to vacuum conditions to allow extrapolation of vacuum Isp from the 3:1 nozzle to an 80:1 nozzle. Using this methodology, the extrapolated 80:1 vacuum Isp is annotated on the plots of 3:1 vacuum Isp for Injector 1 (Test 185) and Injector 2 (Test 229) in Figure 7. Injector 1 (Test 185) extrapolated 80:1 Isp was greater than 320 sec at an MR of 3.0 and Injector 2 (Test 229) was greater than 317 sec at an MR of 2.5. Aerojet's performance tests were conducted at 60 sec to ensure complete chill-in of the system and valid performance readings.

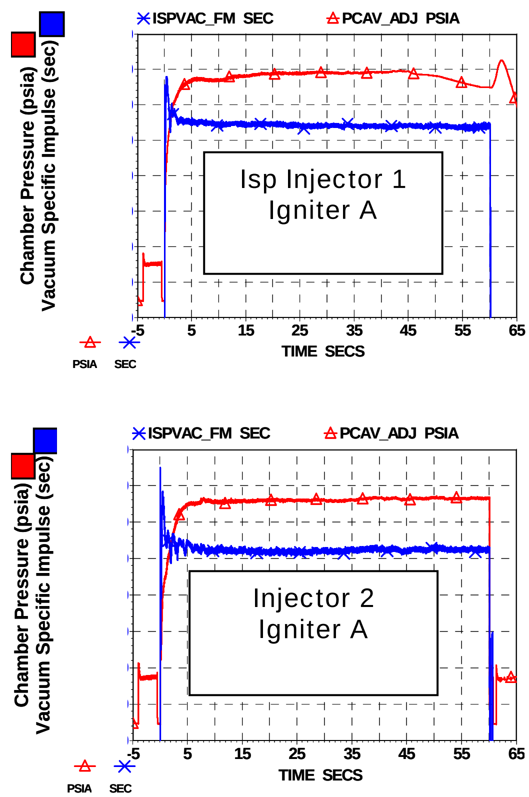

Fig. 7: Measured (3:1) and Extrapolated (80:1) Vacuum Isp for Injectors 1 and 2

MIB and Pulse-to-Pulse Repeatability

Critical maneuvers and on-orbit control are operations that involve the vehicle attitude control system and specifically the RCEs. The maneuvering during critical rendezvous, docking, and station-keeping functions can have large effects on overall vehicle propellant usage and ultimately the required mass of propellant for the mission. RCEs having impulse bit limitations may require null firing or multiple RCE firings to maintain a specific orbit, leading to inefficient propellant usage. A low value for the MIB is

critical for minimizing propellant consumption per unit time since the propellant consumption per unit time is directly proportional to the square of the MIB. Therefore, the critical parameter for finely controlled in-space docking maneuvers and on-orbit station-keeping is a short and repeatable MIB.

The MIB is the impulse bit attained for the minimum EPW that a given engine/valve combination can provide. MIB is a function of several competing design issues including injector dribble volume, engine transient response (primarily chill-in), and valve opening and closing response times. The true impulse bit is the area under the thrust curve that extends from when the valves are completely open to when chamber pressure (Pc) reaches 10% of the rated Pc in the tail-off region of the pulse. Sometimes a simplified method of calculating the impulse bit is used wherein the time period is taken as the EPW, that is, FS1 to FS2, and multiplied by an average thrust over the pulse. However, this simplification does not account for the incremental impulse in the tail-off region, which can be significant for short EPWs. The RCE program has conducted the MIB analysis using the entire area under the curve to where the Pc reaches the 10% threshold after shutdown.

The 100-lbf LO2/LCH4 RCE achieved a remarkable 40 msec EPW operational pulse capability and MIB of < 4 lbf-sec (<17.8 N-sec). Figure 8 shows the pulse-to-pulse repeatability of Injectors 1 (Test 153) and 2 (Test 207) at the 40 msec design limit. Aerojet did achieve impulse bits less the 4 lbf-sec on duty cycles less than 25% and was able to achieve the 4 lbf-sec at the 25% duty cycle using the 40 msec EPW. As expected for such short EPWs, the engine thrust measurement system shows ragged response to the thrust input as indicated by the FA plot in Figure 8. Differences in Pc and FA for Injectors 1 (Test 153) and 2 (Test 207) are due to the different inlet temperature of the propellants for each test. Figure 9 shows the corresponding injector manifold temperatures for these two tests.

Figure 10 shows the MIB and pulse-to-pulse repeatability for EPWs of 160, 80, 60, 50, and 40 msec at a 25% duty cycle. A complete series of MIBs were analyzed for various EPWs—all at a 25% duty cycle. Tests were also conducted at various other duty cycles, for example, at 60 msec EPW/40 msec ECW, to address duty cycles over 50%, as well as duty cycles of 15, 10, and 5%. Successful testing at these duty cycles demonstrates the engine is not sensitive to off pulsing and other operational limitations that would add complexity to the propulsion system operational requirements.

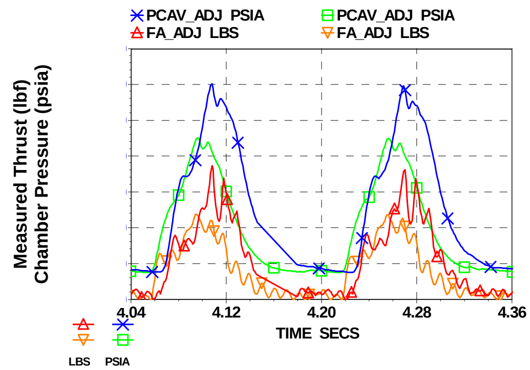

Fig. 8: Injectors 1 and 2 Thrust (Channel A) and Chamber Pressure for EPW of 40 msec and ECW of 120 msec (Test 153/207)

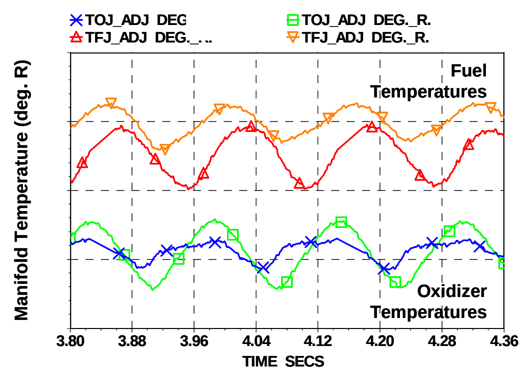

Fig. 9: Injector Manifold Temperatures for Tests 153 and 207 (Injectors 1 and 2)

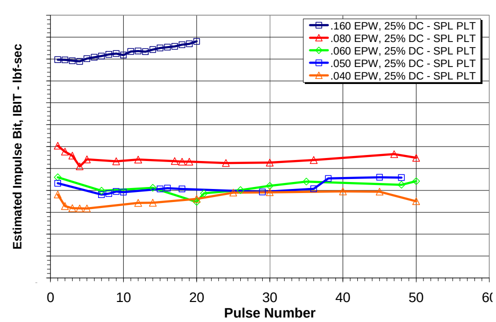

Fig. 10: Minimum IBIT for 25% Duty Cycle at Various EPWs

High MR Operation

The RCE was required to act as a redundant engine for the main engine. To meet this requirement, the system would need the RCE to be capable of operating at MRs of up to 3.5 to match the main engine propellant consumption. If the RCE operates at a lower MR than the main engine, additional fuel would have to be stored on the vehicle to meet this redundancy requirement. This requirement produced considerable risks for the 100-lbf engine and columbium chamber materials traditionally used for a RCE.

Aerojet again chose to innovate with unique injector designs to allow operation of the engine with these MRs and used our virtual injector design computational fluid dynamics (CFD) capabilities to develop two injector patterns to demonstrate the capability. The efforts were focused on Injector Pattern 1 for highest performance that would be used for the vehicle in the +/- Y and +/- Z attitude control and the + X facing RCE could be a separate Injector Pattern 2 specifically capable of pulsing the same as Injector 1 but capable of operating at higher MRs. Injector 2 was developed to explore the limits of the capabilities for high MR operation. Injector 1 was fired up to MR 3.0 and had columbium chamber temperatures of approximately 2300 °F (1533 K) and Injector 2 achieved MR operation greater than 3.6 with the highest chamber temperatures of approximately 2100 °F (1422 K). Figure 11 summarizes the MR and Pc map for Injector 1 and Injector 2 for all tests conducted during Options 1, 2, and 3.

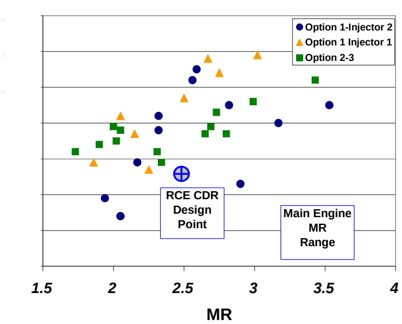

Fig. 11: MR and Pc Operational Map Determined for 100-lbf LO2/LCH4 RCE

During the entire test program, Aerojet used special instrumentation to detect combustion stability modes and our sample rates were in excess of 2.5 times the highest anticipated chamber frequencies. Also, no instabilities were detected for either injector during the entire test program including all gaseous and liquid inlet temperature mapping.

CONCLUSIONS

The NASA/Aerojet team addressed the engine development challenges for a LO2/LCH4 RCE rather than push the risks and complexities back onto the propulsion system. This approach identified the five major challenges/risks that needed to be mitigated for the RCE and developed a detailed development plan that was implemented during the four phases of the program: Basic, Option 1, Option 2, and Option 3. The accomplishments that mitigated the five major risks were:

(1) Developed a passive low MR start transient capability allowing simultaneous valve openings to decrease the complexity and mass of the propulsion system operational architecture

(2) Demonstrated a large igniter inlet temperature map well in excess of the prescribed specification limits

(3) Evaluated two injector designs both with extrapolated vacuum Isp >317 sec, at 80% Bell and a 80:1 nozzle, with one optimized for higher Isp and the other for higher MR operations

(4) Demonstrated an MIB of less than 4 lbf-sec (17.8 N-sec) and a minimum EPW less than 80 msec

(5) Developed a lightweight engine using an integral igniter and injector with an integral exciter and spark plug design to decrease volume and mass while enhancing safety and reliability.

Our Option 1 testing validated the design innovations for the igniter, injector, and overall engine assembly. These innovations produced a lightweight robust engine capable of meeting the propulsion requirements established by the RCE program. This effort demonstrated that an in-space cryogenic propulsion system could be developed that would include RCS engines operating on

cryogenic liquid propellants, and share common tanks and manifolds as the main engine.

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